NACA

RESEARCH MEMORANDUM

HEAT TRANSFER TO BODIES AT ANGLES OF ATTACK

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SUMMARY

Heat-transfer rates were measured on a modified Karman nose shape at an angle of attack of 0° at Mach numbers of 6.86 and 3.69, and at angles of attack up to 25° at a Mach number of 3.69. Data are presented for a smooth model, which showed natural transition, and for the model with roughness strips, which caused fully turbulent flow.

INTRODUCTION

The heat transfer to bodies of revolution at zero angle of attack with laminar flow has been extensively studied, both theoretically and experimentally, so that in general calculations can be made for a given configuration with confidence. For turbulent boundary layers at zero angle of attack, the solutions for cones and flat plates have been used with the local flow conditions along the body to obtain good approximations. Very little study has been made of the heat transfer to bodies at angles of attack. At very large angles of attack, it might be expected that the results found for swept infinite cylinders would be applicable, but there is a decided lack of information on heat transfer at intermediate angles of attack.

This paper presents some of the results from an experimental study of the heat-transfer rates to a particular body of revolution at angles of attack up to 25°, in order to give some idea of the distribution of heat-transfer rates that can be expected at supersonic speeds on an airplane fuselage.

SYMBOLS

N_{St} Stanton number based on free-stream air properties

R_D Reynolds number based on body maximum diameter and free-stream air properties

X axial distance measured from nose

D body maximum diameter

Λ sweep angle, deg

α angle of attack, deg

M Mach number

APPARATUS

The body tested is shown at the top of figure 1. It is a Karman se shape of fineness ratio 5, modified by the addition of a tangent half-angle cone in front, and a length of cylindrical section behind. Tests were made at zero angle of attack in the Langley 11-inch hypersonic tunnel at a Mach number of 6.86, and at angles of attack from 0° to 25° in the Langley Unitary Plan wind tunnel at a Mach number of 3.69.

RESULTS

In figure 1 are shown the results of the tests at the two Mach numbers at zero angle of attack for laminar flow. The laminar correlating parameter $N_{St}\sqrt{R_D}$ is plotted on a logarithmic scale against the axial station along the model in diameters, X/D. The circles are the data from the tests at $M = 6.86$ and the squares at $M = 3.69$. The dashed curves, calculated by the theory of Stine and Wanlass (ref. 1) for the two Mach numbers, fit the data well over the body, except that, at axial stations behind $X/D = 4.2$ at $M = 3.69$, the rise in the data indicates transition to turbulent flow. Also shown in figure 1 are curves calculated for $M = 6.86$, by the flat-plate theory of Van Driest (ref. 2) for the cone and flat plate. The calculated curves for the cone and flat plate agree fairly well with the experimental data on the parts of the body that are nearly conical and cylindrical, respectively.

In figure 2 are shown data for the same model at $M = 3.69$ and zero angle of attack but with turbulent flow tripped by roughness applied in a ring near the nose as indicated in the sketch at the top of the figure. The same parameter $N_{St}\sqrt{R_D}$ is presented in order to facilitate comparison with the laminar values in figure 1. The data near the nose are not quite up to the values calculated by turbulent cone theory (ref. 3) but show the same trend along the length as did the laminar data, and agree well with the turbulent flat-plate theory (ref. 4) toward the rear of the body.

Figure 3 shows the variation of the Stanton number based on the free-stream conditions along the lower or windward meridian line of the body at several angles of attack. On the left are shown the data for the smooth model. The curve at zero angle of attack is the same as was shown in figure 1, with transition starting at $X/D = 4.2$. As the angle of attack increases, the heat-transfer rates increase, as would be expected, and the point of transition moves forward, up to 14° angle of attack. The curves at angles of attack of 21° and 25° show a somewhat different trend.

In order to obtain turbulent heat-transfer data, a strip of roughness was applied in a band along the lower meridian line as well as in a ring near the nose. The results are shown on the right in figure 3. As in the case of laminar flow, the Stanton numbers increase with angle of attack from 0° to 14°. The curve for $\alpha = 21^\circ$ does not follow the trend with angle of attack, and is in fact lower than that for $\alpha = 14^\circ$. The curve for $\alpha = 25^\circ$ was very irregular and will be discussed later. The behavior of the Stanton number distribution curves suggests that the flow around the body changes to a mainly crosswise flow at some angle between 14° and 21°.

In figure 4 the distribution of Stanton number along three meridian lines is shown for two angles of attack, 7° on the left, and 25° on the right. Again, roughness was applied to produce turbulent flow. At an angle of attack of 7° the Stanton number drops from its value at the windward meridian to about half at 90°, and about 1/3 on the leeward side (180°). On the leeward side, behind $X/D = 3$, the Stanton number is independent of X/D, so that separated flow is indicated.

At an angle of attack of 25°, the values on the windward meridian are very erratic, ranging from values near those found for laminar flow up to values in good agreement with the turbulent swept-cylinder theory. This behavior is due to the fact that the local Reynolds numbers over the roughness strip are very low, so that transition is not fully completed at all stations. The range of values of the data represents various stages in the transition to fully turbulent flows. The curves at meridian stations 90° and 180° are smooth. Values of Stanton number at the 90° station for $\alpha = 25^\circ$ are not very different from those found for $\alpha = 7^\circ$, but at the 180° station are somewhat lower than at $\alpha = 7^\circ$.

Figure 5 shows a polar plot of Stanton number around the circumference of the body at $X/D = 5.12$, where the body is cylindrical, for several angles of attack. On the right side are the distributions for fully turbulent flows, produced by a roughness strip along the windward meridian. On the left side of this figure are values obtained on the smooth model. As the angle of attack increases, the fully turbulent Stanton numbers increase on the windward side and decrease on the leeward side up to 14°. The curve for $\alpha = 21^\circ$ nearly coincides with that

for $\alpha = 14^\circ$, and the curve for $\alpha = 25^\circ$ shows a further increase. The long-short-dash curve for $\alpha = 25^\circ$ is calculated by a semiempirical method developed for swept cylinders. The basis of the calculation is the theory for the average heat-transfer coefficients over the front half of swept circular cylinders with fully turbulent boundary layers presented by Beckwith and Gallagher in reference 5. This theory was combined with an unpublished experimentally measured distribution of local heat-transfer coefficients around a swept cylinder obtained at the Langley Laboratory by Beckwith and Gallagher, on the assumption that the distribution curve is independent of sweep angle and Mach number (as has been shown for laminar flow). The agreement with experiment is very good.

On the left side of the figure are shown the distributions of Stanton number obtained on the smooth model at $\alpha = 0^\circ$, 7°, and 25°. The values at $\alpha = 0^\circ$, with natural transition, are in fair agreement with the artificial transition values on the right. At an angle of attack of 7°, the boundary layer is laminar near the windward meridian and low heat-transfer values are obtained. Transition starts at a meridian angle about 45°. At $\alpha = 25^\circ$, transition starts earlier, about 20° from the lower meridian line, after which the Stanton numbers increase and fair into the curve for fully turbulent flow at $\alpha = 25^\circ$ which was transposed from the data on the right.

In figure 5 the data at $\alpha = 25^\circ$ agree well with crossflow theory. The behavior of the curves at $\alpha = 14^\circ$ and 21° indicates that the type of flow may be changing in the interval between. To show the trend more clearly, the Stanton numbers on the windward meridian are plotted in figure 6 against angle of attack for two stations on the model; $X/D = 2.8$ and 5.1. The curve labeled "longitudinal-flow theory" was calculated by flat-plate theory, by use of the local flow conditions outside the boundary layer for the equivalent free stream. The curve labeled "crossflow theory" was calculated by use of the semiempirical method described earlier for swept cylinders, the body being considered to be made up of a series of cylinders of varying diameter and sweep angle. The experimental values increase uniformly from $\alpha = 0^\circ$ to 14° but the trend does not agree with that predicted by the modified flat-plate (longitudinal-flow) theory. At $\alpha = 21^\circ$ and 25°, the data agree well with the crossflow theory curve.

CONCLUDING REMARKS

The experimental results presented in this paper have shown that at zero angle of attack, the available theoretical methods give a good estimate of the heat-transfer coefficients on a body of revolution. At large angles of attack, the methods based on the cross flow over swept

cylinders are applicable. For intermediate angles of attack, however, there is as yet no theoretical approach, and reliance will have to be placed on extrapolation from experimental studies. It is believed that the results presented in this paper can be considered typical of simple fuselages of moderate fineness ratio at supersonic Mach numbers for which the air may still be considered an ideal gas.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., March 6, 1957.

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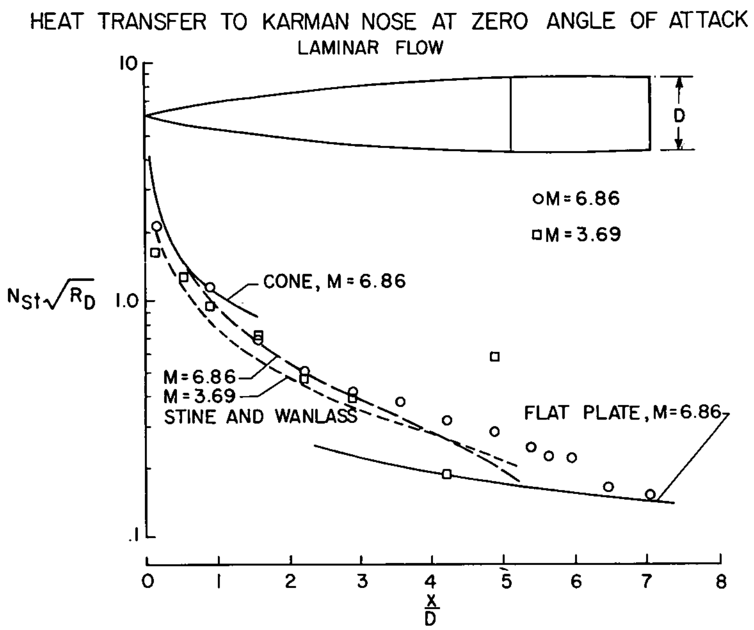

Figure 1

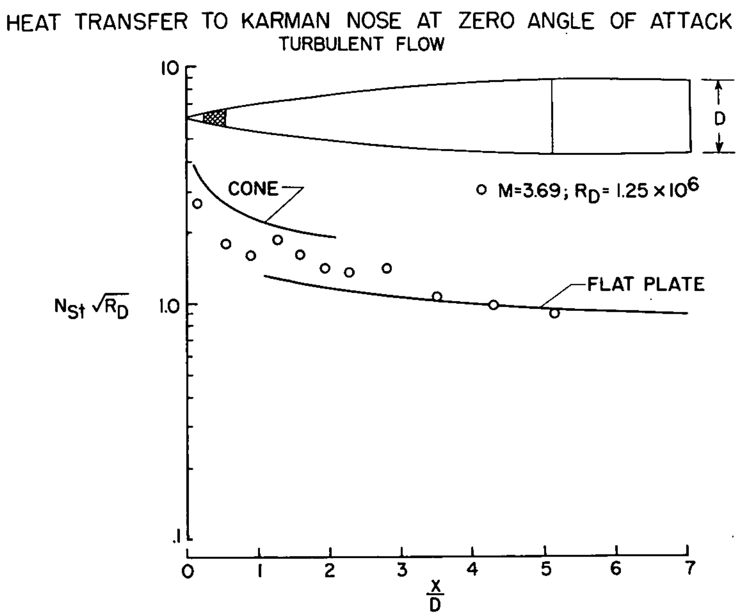

Figure 2

HEAT TRANSFER TO LOWER MERIDIAN

KARMAN NOSE; M = 3.69; $R_D = 1.2 \times 10^6$

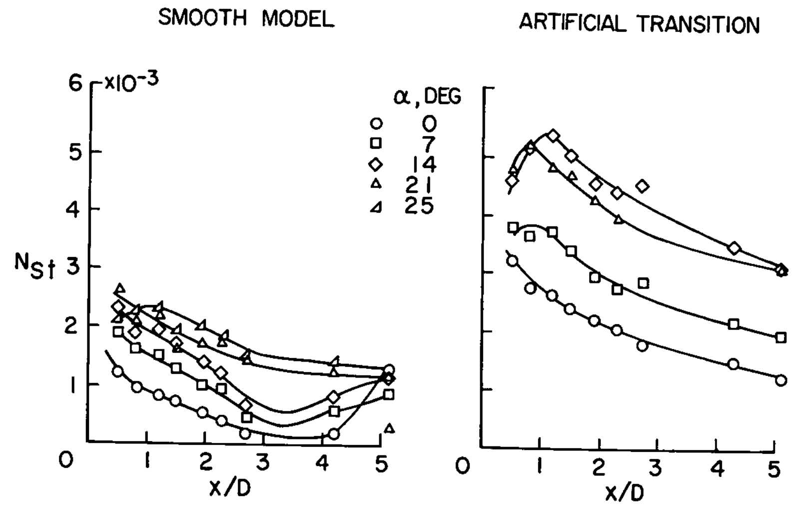

Figure 3

HEAT TRANSFER TO KARMAN NOSE

M = 3.69; $R_D = 1.2 \times 10^6$; ARTIFICIAL TRANSITION

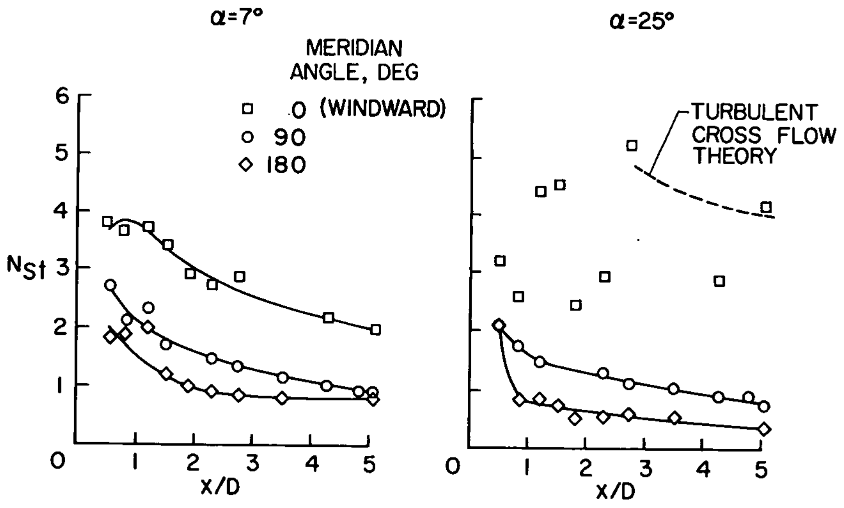

Figure 4

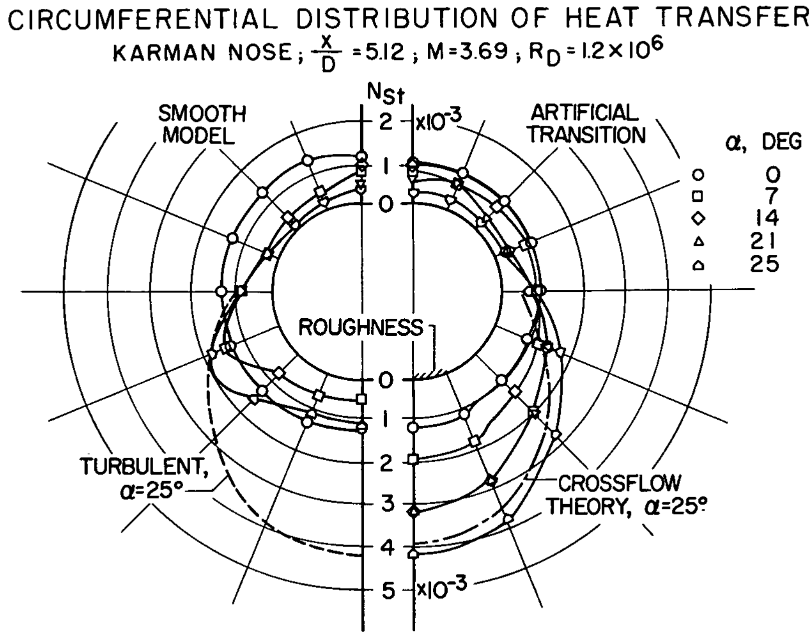

Figure 5

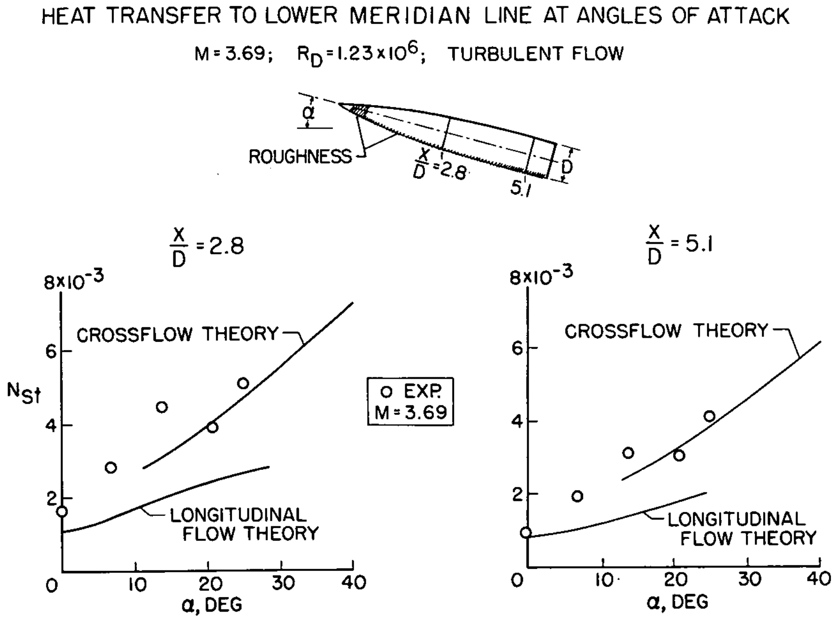

Figure 6